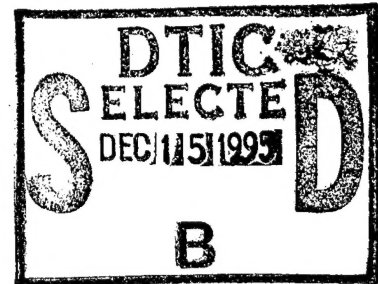


## Power Subsystem Trade Studies for a 5 kW MEO Application

1 December 1995



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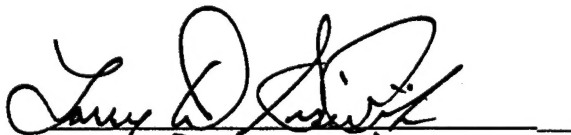
**THE AEROSPACE  
CORPORATION**

El Segundo, California

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This technical report has been reviewed and is approved for publication. Publication of this report does not constitute Air Force approval of the report's findings or conclusions. It is published only for the exchange and stimulation of ideas.

A handwritten signature in black ink, appearing to read 'Larry D. Sirwaitis', is written over a horizontal line.

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## Executive Summary

The system trades undertaken for a large, generic mid-earth orbit (MEO) (~10,000 km) spacecraft have been studied. The objective of the study was to address the main factors related to a modern, but not cutting edge, spacecraft power system capable of acceptable reliability over a 10-year mission. The spacecraft was assumed to have onboard control functions with ground control as a backup. Where feasible, hardware items were recommended and manufacturers identified that already have an established ground-based data base, and where applicable, also have an established flight history. Following a review of the issues and trades, recommendations were presented, along with supporting data. In this Executive Summary, only the results are presented.

A typical MEO orbit mission passes through the earth's radiation belts and exposes the spacecraft to high levels of natural radiation. This affects the power system in two ways: (1) the solar array must be large and heavy, and (2) the power system electronics (as well as other on-board electronic systems) must use radiation-hard parts. Radiation-hard electronics are expensive. For proper mission optimization, the benefits of operating the spacecraft in the orbit chosen must be weighed against the disadvantages caused by the high level of radiation experienced by the spacecraft in this orbit. Mission optimization was not performed as part of this study, although The Aerospace Corporation regularly performs this task for U.S. Air Force satellite programs.

The solar array design is driven by the high levels of natural radiation that are present in the assigned orbit. We recommend the use of commercially available gallium arsenide solar cells that have been grown on 5.5 mil germanium substrates. The array is recommended to be a rigid panel design, which provides both support and backside shielding for the solar cells. The cover glass shielding the topside of the cell should be thick. Our calculations using typical values for substrate density and electrical losses indicate that a cover shielding thickness of 27 mils of equivalent fused silica would provide the optimum specific power for the array. The array area of this design that is required to provide 5.0 kW at the end of the 10-year mission in the designated

orbit is 34 m<sup>2</sup> (vs 66.5 m<sup>2</sup> for an array populated with silicon solar cells). The load power at the end of life (EOL) for this spacecraft is assumed to be 5.0 kW. Given the assumptions stated in the text, the required array power at EOL is predicted to be 6.68 kW. This power level will depend on array pointing and spacecraft load profile. An array populated with GaAs/Ge solar cells having a total panel area of 45 m<sup>2</sup> would be required to provide this EOL power of 6.68 kW, and it would weigh about 207 kg at the panel level. A higher efficiency solar cell using multiple bandgaps would reduce the area and weight required. This type of cell is in development and could become available at a usable cost within 3 to 5 years.

In the battery portion of the power subsystem, maximum advantage was taken of the present experience related to the design, manufacturing, storage, and degradation issues associated with state-of-the-art nickel hydrogen cells. The power system architecture and thermal control aspects of the spacecraft are known to have significant impacts on the cycle life and performance of the battery. These issues will be mentioned where they might affect the battery life, but this short study will not address them in detail. We recommend state-of-the-art, individual pressure vessel (IPV) nickel hydrogen cells assembled into redundant 22-cell batteries without the use of bypass diodes for the individual cells. If higher voltages are required by the spacecraft, we recommend the use of boost converters. Maximum use should be made of modern battery storage and management procedures. This would include the use of majority voting strain gauges to measure pressure and majority voting thermistors to measure temperature. These circuits, constructed of electronic circuits that would resist the natural radiation encountered at this altitude, would be used to control the battery recharge and reconditioning functions.

For the application under consideration, a typical battery system would be as follows. There would be four parallel batteries, each with a nameplate capacity of 71 Ah (85.2 Ah actual capacity), cycling to 50% depth of discharge (DOD) based on the nameplate capacity. With the loss of one battery, the DOD would be increased to 66%. There should be individual charge controllers for each battery, which will allow occasional reconditioning of the batteries and optimum battery management with a degraded battery. At a DOD of 66%, the cycle life requirements of the mission would be met using cell designs that already exist and have been

validated in life cycle testing. To enhance the cycle life of the batteries, recharge ratios must be set to minimize the amount of overcharge without allowing any of the cells to progressively run down in capacity during cycling. Onboard computers using ampere-hour integration in conjunction with majority voting of strain gauges to measure and monitor the pressure should be employed to manage battery recharge. The four batteries would weigh about 57.8 kg each, for a total of 231 kg. This weight includes the mounting fixtures, baseplate, and wiring, but excludes the heat rejection subsystem. Due to the importance of the heat rejection subsystem in minimizing the temperature variations within a battery, we recommend the use of centrally located thermal flanges to conduct the waste heat to the cold plate and heat pipes to transfer the heat to a space radiator. This should limit the temperature swings to within  $3^{\circ}\text{C}$  to  $5^{\circ}\text{C}$  of the nominal operating temperature of  $+5^{\circ}\text{C}$ .

Details covering the basis for the various recommendations and sample calculations are included in the following sections of the report. Section 1 describes how this report has been prepared and how the results will be presented. Section 2 reviews the different cell and battery options. Section 3 considers the spacecraft architecture, redundancy, and inflight management issues. Section 4 reviews the solar cell and array options. Section 5 summarizes the recommendations for the mission under study.



## 1. Introduction

The following report will address the requirements of a large, generic mid-earth orbit (MEO) spacecraft orbiting at about 10,000 km. In particular, the power system will be considered, and the various options related to selecting individual pressure vessel (IPV) nickel hydrogen cells for the energy storage function will be reviewed. Following a brief discussion of each of the trades to be considered, recommendations will be presented, based on a 5.0 kW average load. These recommendations will, at times, be interrelated with the suggested overall power system, the spacecraft's architecture, the redundancy philosophy, and the on-orbit operation of the spacecraft. Therefore, issues related to solar array selection, spacecraft design/operation, or any other factors that may impact the batteries operational capabilities will be included.

These recommendations will be based on the authors' role of dealing with power system issues, ranging from the fundamental understanding of electrochemical and photovoltaic processes associated with the individual components, to assistance in the on-orbit operations of spacecraft experiencing operational difficulties. As such, staff members have worked with cell and battery manufacturers on a confidential basis, assisting them in the diagnosis and resolution of solar cell and battery-related issues during the design, manufacture, and storage stages of these flight articles. Over several decades, staff members have developed a large number of unique analytical techniques, which have led to the resolution of these issues. Many of these procedures have been documented and published. Where helpful references are available, they will be cited.

This ongoing engineering support applied to power system issues has been very helpful to the many programs we have supported and has established Aerospace as a recognized leader in this unique field.

## 2. Cell and Battery Technology

### 2.1 Status of Nickel Hydrogen Technology and Cycle Life Testing

Nickel hydrogen technology is newer than nickel cadmium technology. Following experimental flights in 1976 and 1977, it began service in 1983 on the Intelsat 5-B series of geosynchronous earth orbit (GEO) communication satellites. Since then, there has been a gradual shift away from nickel cadmium to nickel hydrogen by many newer flight programs. This has been due to the higher usable energy density at an equivalent life for certain missions using this newer technology. The launch of the Hubble Space Telescope in 1990 was the first major use of nickel hydrogen in a low earth orbit (LEO) application. Both of these applications have used IPV cells. In this configuration, the components for only one cell are placed within a pressure vessel. The energy density of LEO and GEO designs in the range of capacities from 50 to 90 Ah varies from about 50 to 60 Wh/kg at the cell level. Many design variations within the IPV family of configurations have been manufactured, tested, and flown. As a result of the large volume of cells manufactured and 20 years of experience with this technology, a number of manufacturing, storage, and use-related issues have already been resolved, and most of the idiosyncrasies are reasonably well understood.

Nickel hydrogen technology was originally built up around what is called the 3.5 in. diameter hardware. To a lesser extent, cell diameters of 2.5, 4.5, and 6 in. have been successfully built, tested, and flown. Eagle Picher, Gates Aerospace Batteries, Hughes Aircraft Company, Yardney, and SAFT have all produced several of these cell types. Recently, Yardney withdrew from nickel hydrogen cell production, and Gates Aerospace Batteries was purchased by the French battery manufacturer, SAFT. The 3.5 in. refers to the nominal diameter of the thin-walled Inconel 718 cylindrical section of the pressure vessel used to contain the high pressure hydrogen gas. The capacity of this type of cell is varied by stacking the required number of positive and negative plates onto a centrally located spool piece. Within certain limits set by the hydroforming process to deep draw the Inconel into a cylinder with a hemispherical shaped end, the capacity can be varied over a wide range. From an energy density point of view, however, there are capacity

boundaries that would suggest different diameters than the basic 3.5 in. Computer codes have been established that are used to suggest the optimum diameter to yield the highest energy density. The industry has adopted 3.5 and 4.5 in. as the two diameters that allow cells to be produced with capacities that cover the range of 25 to 250 Ah. Within these cell types, designs that emphasize cycle life, pulsing power, or high energy density have been investigated on an experimental basis. Optimization of one of these three factors usually results in a performance trade-off in at least one of the other factors. The different cell manufacturers are best able to describe the attributes of their product line.

Life cycle testing programs have been in place for many years to develop the data base to establish the confidence in projecting the usable life and reliability of the energy storage portion of a spacecraft's power system. Figure 1 presents data from the Air Force funded life cycle testing being carried out at the Navy testing facility located at Crane, Indiana. Some recent data from

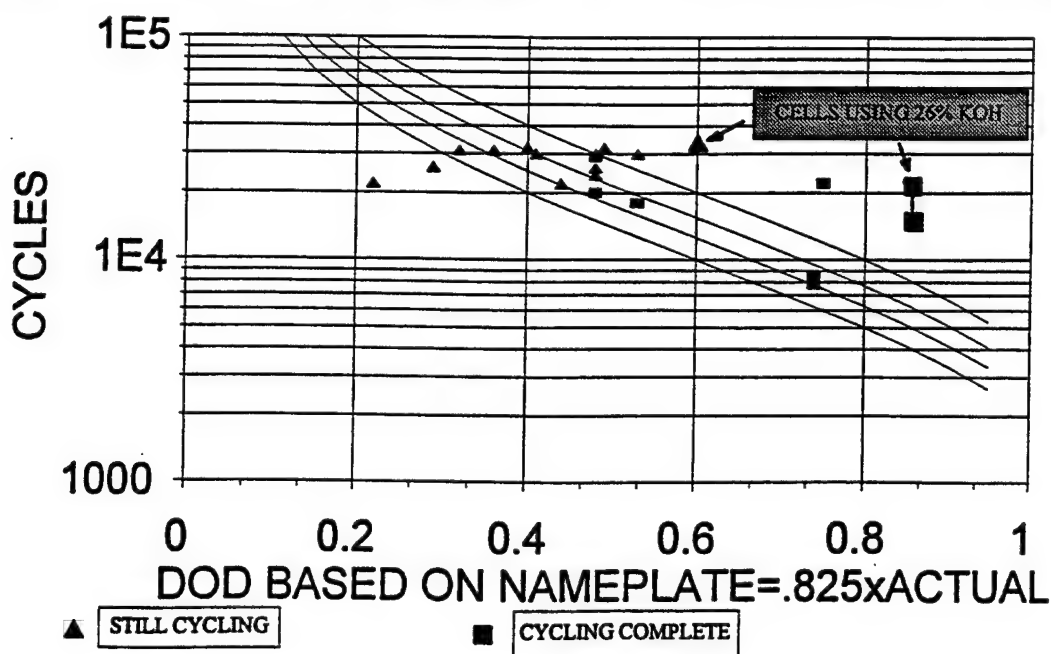


Figure 1. Cycle life vs depth of discharge.

NASA-funded tests are also contained in this figure. A testing program that also has produced encouraging cycle life data is being carried out at Martin Marietta in the Denver area. The most recent compilation of data presented in a manner suggesting a relationship between cycle life and

depth of discharge (DOD) appears in reference 1. Selected NASA data have been updated via private communication. Most of these data points are for tests of more than one cell that are still continuing. Where tests have been completed that support the case for the recommended DODs, they will be included as well. These tests suggest that cycle lives in the tens of thousands are achievable at 66% DOD. One data set that is particularly encouraging is the 32,000 cycles at 60% DOD testing on a group of three cells having advanced design features. These cells were built to NASA Lewis Research Center's specifications by Eagle Picher (Joplin, Missouri).

## 2.2 Component Selection Trades

There are a number of choices related to the components that could be selected for a modern nickel hydrogen cell. These include choices between dry powder and slurry plaque substrate material for the positive electrodes, the impregnation technique, the loading level of active material within the electrode plates, the amount of cobalt additive, and the type of separator material. In the following paragraphs, these topics will be reviewed in light of the latest understanding of the cell electrochemistry and current status of cycle life testing. The cycle life requirements for this application, about 14,600 cycles over a 10-year period, are considered to be relatively stressful. For this reason, the recommendations in regard to component selection will take these requirements into account. It should be noted that not recommending a particular variant does not necessarily mean that it would not perform satisfactorily. Where concerns or cautions do exist, they will be mentioned, along with our explanations.

### 2.2.1 Electrolyte Concentration

Using 26% KOH as the electrolyte, vs the more traditional 31%, has resulted in significant increases in cycle life in flight-weight cells at the expense of a small decrease in energy density. The reduction in electrode swelling appears to be the main factor in this increased cycle life at deeper DODs. The lower electrolyte concentration favors the beta-beta nickel electrode reaction, whereas the higher concentration promotes the alpha-gamma electrode reaction with its significantly higher density changes over the course of a complete cycle. Since cycle life

requirements for this application are moderately severe, the use of 26% KOH is recommended. Higher electrolyte concentrations have been used in other programs, but because of the high DODs to be recommended in later sections and the required cycle life, we would recommend against the use of 36% KOH. Possibly a case could be made for 31% KOH, but 26% is recommended based on the available life cycle testing (Refs. 2, 3, 4).

### 2.2.2 Separator Material

Adequate performance and life of nickel hydrogen cells in GEO and LEO life cycle testing have been obtained using properly prepared asbestos separator material. However, other factors strongly suggest that asbestos separators should not be used in this application. These factors include assurance of a continuing supply of asbestos and the environmental hazards associated with its processing and use. Compared with two layers of 0.012-0.014 in. Zircar, the operating voltages are about 20 mV lower when asbestos is used. The quantity of electrolyte retained in cells using asbestos as the separator is typically about 3.0 g/Ah of nameplate capacity. This amount is not considered appropriate for this application. Further, it has been reported that certain naturally occurring impurities within the asbestos have resulted in severe performance problems in completed cells (Ref. 5).

It is known that SAFT has successfully used a polyamide separator material for many years. This results in a lighter cell since the nonwoven polyamide material is considerably lighter and retains less electrolyte than either one or two layers of Zircar. There are several reasons for not being able to recommend the use of this lighter weight material. The work reported by Lim et al. (Ref. 6) and verified by Zimmerman (Ref. 7) outlines the degradation that takes place with this type of material as a result of oxidation by oxygen and hydrolysis by potassium hydroxide. These reactions are accelerated at elevated temperatures. From past history with other spacecraft, the operational temperatures of the batteries have not always turned out as predicted by designers. The use of the more robust characteristics of the ceramic Zircar material guards against any eventuality in this regard. Further, the ability of Zircar to hold more electrolyte compared to asbestos or polyamide materials facilitates the maintenance of the electrolyte distribution that is

needed for proper cell operation. We cannot recommend the use of polyamide based materials for this application.

Other materials (Ref. 8) are currently being evaluated in experimental cells as potential lower cost, lower weight alternatives to Zircar. These tests have not yet reached the level of maturity that would allow us to recommend their use at this time. Flight-type cells have been built and are being successfully tested with one layer of Zircar. This design is not considered robust due to the significant reduction in quantity of electrolyte. We recommend the use of two layers of Zircar for this application because of its established test and flight history.

### 2.2.3 Plaque Type

Slurry plaque has been used since the late 1930s following its development in Europe. The dry powder process is a more recent development and is used for aerospace applications by several manufacturers. The dry powder process appears to be more labor intensive than the slurry process. However, it has been reported that a more uniform distribution of pore sizes (Refs. 9, 10) results from the dry powder process. It is generally accepted that the slurry plaque material is less porous (~80% vs ~84%) but stronger than plaque made using the dry powder process. Both of these processes sometimes experience quality control problems. Special tests have been developed to augment manufacturing acceptance tests where there are quality control issues. These tests are outlined in more detail in the section on Manufacturing Concerns. For a LEO mission, we might recommend a high quality slurry material due to the rigors of that application. For the application at hand, we recommend the use of dry powder plaque material.

### 2.2.4 Impregnation Method

The chemical impregnation method is generally not used to make plate material for nickel hydrogen cells. Rather, both the "aqueous" and "alcoholic" processes for electrochemical impregnation have found general acceptance. Each manufacturer has his own favored process. For the application under consideration here, we do not have a recommendation other than to mention that both processes are subject to quality issues from time to time. Depending on the

selected vendor, either one or the other impregnation process will be in place. Following some early studies (Refs. 9, 11), a value of about 1.6 g of active material per cubic centimeter of void volume has been commonly used by cell manufacturers. Here again, special tests have been developed by Aerospace to help screen out finished plate material that has the proper amount of loading from a weight standpoint, but has an unacceptable distribution of active material within the pores. When utilized with the manufacturer's acceptance test data for nickel electrodes, these special screening tests will ensure uniform utilization and capacity from one manufacturing lot to another. The level of cobalt or other additive used in the impregnation step is generally considered proprietary, and direct discussions with the vendor are recommended to review the suggested attributes of the type and level of additive used. It is well documented that the higher amounts of cobalt (10% vs 5%) generally result in a higher conductivity of the active material and thus result in slightly better performance (Ref. 12). However, electrodes containing 10% cobalt do appear to be more sensitive to various storage related capacity fading issues. Both the aqueous and alcoholic processes have produced plate material that has fully demonstrated the cycle life requirements of this application.

#### 2.2.5 Terminal Seal Type

There are three types of terminal seals in common use: molded Nylon, compressed Teflon, and ceramic bushing. All three have been found to perform satisfactorily when properly fabricated. At times, each of these types of seals has been a source of difficulty. Each manufacturer has a preferred terminal type, and the terminal type is typically not an option available to the customer. A 5/8 in. terminal size is recommended due to the probable high rates of discharge for this application.

### 2.3 Cell Design Considerations

There are literally thousands of variables in cell design. The main variables, which will be given in the following paragraphs, are stacking arrangement, cell end of charge pressure, precharge amount and type, and terminal arrangement.

### 2.3.1 Stacking Arrangement

Originally the Comsat Corporation developed what came to be known as the "Comsat" design. These cells were "back to back" in terms of their stacking arrangement and had the positive and negative buses located along opposite edges of the stack of plate pairs. The "Air Force" configuration originally used the "recirculating" stacking arrangement of cell components. See Figure 2 for clarification of these differences. The thin nickel foil plate tabs were arranged as a bundle within the central core of the cell, one bundle going up to one terminal, and the other bundle going in the opposite direction to the terminal at the other end of the cell. Both of these

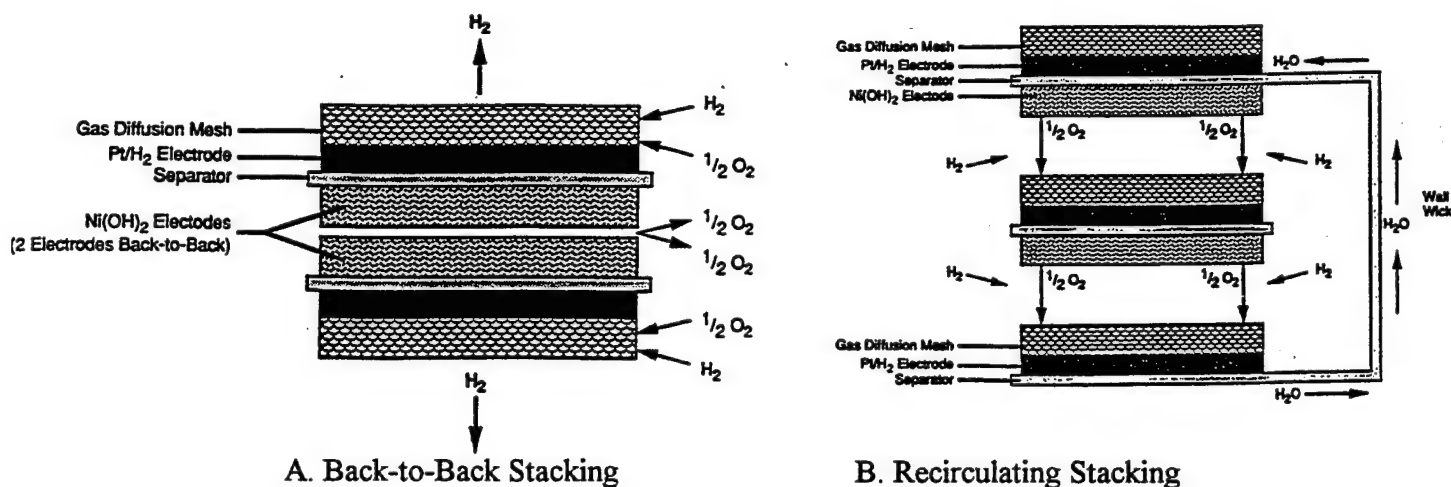


Figure 2. Stacking arrangements for nickel hydrogen cell components.

cell designs have performed satisfactorily. Gradually, it was found that the back-to-back arrangement was better able to handle the oxygen that is generated during the later portions of the recharge cycle. This was particularly true when low bubble pressure Zircar separators were used. The back-to-back configuration was less susceptible to the destructive "popping" damage to the hydrogen electrode. The back-to-back stacking arrangement also resulted in a shorter, lighter weight stack of plate pairs. There are several other factors affecting IPV nickel hydrogen cells that have led us to recommend the use of a modified Air Force design with a Comsat stacking arrangement (usually referred to as the "Man Tech" design). Cell sizes have increased from the 25-35 Ah range in the early 1980s to the 65-100 Ah sizes that are now very common. The large capacities per cell require a center girth weld, with the stack split between the upper and lower



half of the cell. The asbestos, used successfully in the original cell types, was of a variety that either had smaller amounts of detrimental impurities than is currently available or had undergone a "remanufacturing" process that removed these constituents.

The Air Force and Man Tech cell designs have a wall wick to help redistribute the electrolyte throughout the cell. Later versions catalyzed certain portions of the porous ceramic material to encourage the oxygen to recombine with the hydrogen on the catalyzed portions of the wall, as opposed to the more delicate hydrogen electrode. Depending on the rate of charge and the amount of overcharge, the use of catalyzed wall wicks may not be necessary. However, in limited life tests, cells with catalyzed wall wicks have always outlasted equivalent cells without catalyzed wall wicks. We recommend the use of catalyzed wall wicks for this application.

### 2.3.2 Cell End of Charge Pressure

Many cells have been built, tested, and flown with a 650 psia maximum operating pressure at the beginning of life. The cells on NASA's Hubble Space Telescope have been designed to operate up to about 1100 psia. A safety factor of about 3 is designed into both of these cell vessels. The vessels themselves are designed to leak before they burst. A small gain in energy density is seen in the higher pressure versions at the expense of a higher rate of self-discharge. A higher rate of self-discharge translates into the need for a higher recharge ratio and thus more solar array power. Since manufacturers can design cells with different end-of-charge pressures, we recommend that the trade between energy density and solar array requirements be reviewed for the particular mission.

### 2.3.3 Precharge Amount and Type

Nickel hydrogen cells built in the late 1970s through the mid- to late-1980s were designed to have an excess of hydrogen gas in them when fully discharged. This came about because the designers felt that at the higher hydrogen pressures, the cell would work better, and that if there was a low level leak, it would take longer for the cell to become inoperative. These cells

appeared to work fine until extended periods of storage at low cell voltages were experienced. A number of studies confirmed that at low nickel electrode potentials, hydrogen gas could react with the discharged nickel hydroxide to produce entities that resulted in the permanent loss of usable capacity (Refs. 13, 14, 15). It was found that by having excess positive material relative to the amount of hydrogen, this situation could be avoided. This topic will be covered in more detail in the section on recommended storage procedures. At this point, we strongly recommend that the cells be designed to have positive precharge.

#### 2.3.4 Terminal Arrangement

Cells can be built with both terminals at one end (rabbit ear design) or with a terminal at each end (axial design). In the rabbit ear design, the difference in resistance of the tabs due to their different lengths can result in the plates closest to the terminals discharging first. Batteries made with cells using this newer variant require less head space. Cell designs with terminals at both ends will discharge the plates more evenly. For this application, we recommend the axial arrangement for the terminals.

#### 2.4 Manufacturing Concerns

Based on the technical support provided to the Air Force programs by the scientists and engineers of The Aerospace Corporation, we have been intimately involved in the identification and resolution of the following manufacturing issues. As such, we have accumulated a significant data base that allows us to address and help manufacturers avoid a variety of quality and performance problems.

##### 2.4.1 Manufacturing Issues Prior to Assembly

Nickel plaque material is typically specified according to the desired degree of porosity. This number averages about 82%; it is usually lower for plaque made by the slurry process and somewhat higher for plaque made by the powder process. Plaque material can be improperly

made and still have the proper value of overall porosity. Desirable plaque material has a particular pore size distribution, which coincidentally has an associated value for the porosity percentage. However, the reverse is not true. That is, material with a certain overall porosity will not necessarily have the proper pore size distribution. Material that has been sintered improperly (too high a temperature and too high a belt speed, for instance) can result in a product that has a surface layer made up of very fine pore sizes. The interior of this material can be more porous than usual due to being under-sintered such that, on an overall basis, the material has the specified porosity. If this situation is not detected, impregnation or performance problems can result. The technique that has been developed and utilized to detect and measure this and other anomalous sinter conditions is similar to the principle used in the scanning tunneling microscope (STM) (Refs. 10, 16).

Impregnated plaque material is usually called plate. The amount of active material deposited either chemically or electrochemically into the plaque can be specified in terms of grams per cubic centimeter of void or in terms of grams per square centimeter of frontal area of plate. This active material is typically measured according to the weight pickup of plaque material that has a given porosity. Here again, it is generally assumed that the active material has been deposited evenly throughout the interior of the plate. Finished plate is tested for usable capacity per unit area as a way of measuring the suitability of the plate for inclusion into the assembly process for the cell. Problems arise when active material has been disproportionately deposited near the surface of the plate. This type of plate will have very low porosity near the surfaces and thus poor mass transport characteristics. This type of plate generally has poor capacity per unit of frontal area. The technique to verify the existence of this maldistribution is called profiling (Refs. 17,18). By using an ion microprobe analysis across the width of the electrode, the loading profile of active material can be detected. Using this technique, evidence of maldistribution of active material was found in plate material that had been prepared for a NASA program. Both the cobalt and nickel content were determined to verify that it was active material. Scanning electron microscope photos of this same electrode qualitatively confirmed the presence of higher loading at the front and back surface of the electrode.

## 2.4.2 Cell Assembly and Acceptance Problems

The intent of screening tests is to ensure that only quality plaque material that has been converted into properly impregnated plates is assembled into finished cells. Following assembly into cells and the addition of electrolyte, an activation procedure is carried out that is particular to the manufacturer. Following this procedure, cells typically undergo (among many other tests) a 72-hour capacity retention test. The purpose of this test is to identify cells that have low level short circuits. The rate of capacity loss as a function of temperature and hydrogen pressure is reasonably well known. This permits the capacity remaining after the 72-hour stand test to be evaluated against the known rate of self-discharge in the absence of any short circuits. Unfortunately, this test has not been found to be as sensitive as required to exclude cells that have potentially significant short circuits. Reference 19 describes a method (reported by Donley et al.) that permits the accurate measurement of low level short circuits that might otherwise have gone undetected, using the typical capacity retention requirements of a 72-hour stand test. Special statistical analysis of the manufacturer's charge retention test data has been found to indicate when application of the more sensitive test was appropriate.

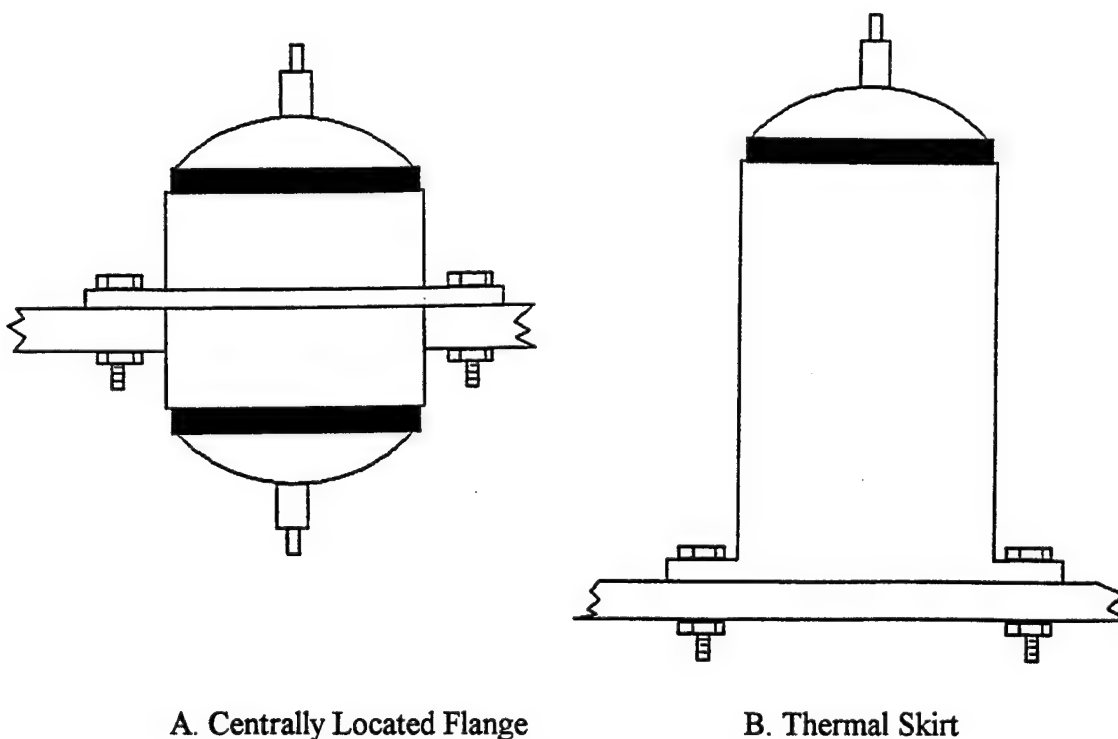
The loss of electrical isolation between the cell case and the internal stack of electrodes will typically result in corrosion of the pressure vessel, and ultimately, in cell failure. This would result in an "open" failure and *must* be avoided. A test based on polarization of the case (the internal surface of the pressure vessel) due to its contact with the cell stack has been developed and successfully used to verify that the cell case is isolated following cell activation. The combination of this test with existing manufacturing checks for detecting short-circuit problems is a key technique for assuring cell reliability and manufacturing quality control.

## 2.5 Battery Design Considerations

### 2.5.1 Attachment to Cold Plate

Cells can be assembled into batteries using thermal skirts, which conduct the heat to a base plate located beneath the cells. Many of the battery designs using cells manufactured by Gates,

Eagle Picher, and SAFT have used this approach. A second design uses a thermal flange located at the midpoint of the cell. This second design is seen on many of the batteries using cells produced by Hughes and some manufactured by Eagle Picher. Figure 3 shows the difference between these two designs. Both designs require the cells to be electrically isolated from the thermal sleeve portion of the battery. In the thermal skirt design, care must be taken to not have an excessive amount of the radiative heat loss from the bottom of the cell to the cold plate. This can result in condensation of water from the electrolyte onto the cold portion of the cell case. This is the so-called cold finger effect, which has been reported by the Comsat group (Refs. 9, 20). The second design, with its thermal flange, has a larger footprint since the cells have a greater spacing between each other. Locating the flange at the midpoint of the cell results in about one-half of the thermal gradient along the length of the cell compared to the thermal skirt configuration. For this application, it is recommended that centrally located thermal flanges be used if there is enough room in the battery compartment of the spacecraft. The thermal skirt design, if used, must preclude the condensation of water vapor at the cell dome that is in radiative view of the cold plate. It is assumed that heat pipes will conduct the waste heat away from the cold plate to the space radiator. The thermal flanges and overall system thermal design must keep temperature gradients measured from the hottest to the coldest portion of the cells below a temperature that would result in excessive water transport within the cell.



A. Centrally Located Flange

B. Thermal Skirt

Figure 3. Attachment options used with nickel hydrogen cells.

### 2.5.2 Cell Operating Temperature

Since nickel hydrogen cells have a higher rate of self-discharge than nickel cadmium cells, and the rate of self-discharge increases rapidly with temperature, it is important to design the battery to operate at a low temperature in keeping with the ability to reject the large amounts of waste heat associated with this large spacecraft. We recommend  $+5^{\circ}\text{C}$  for the mean temperature at the cell's midpoint. We do not recommend a design temperature to exceed  $+10^{\circ}\text{C}$  or one that is below  $0^{\circ}\text{C}$ . The highest heat load will occur during the latter portions of discharge when the discharge current is high. These heat loads can be accurately modeled from thermal design efforts, and should eventually be evaluated in a power system thermal verification test.

### 2.5.3 Use of Strain Gauges and Thermistors

The use of strain gauges and thermistors is strongly recommended. The reliability of these devices has been gradually accepted, resulting in their incorporation into the charge control function of several spacecraft. Onboard ampere-hour integration may be the primary form of charge control, with pressure as another option. Temperature-compensated pressure measurement is an excellent way of monitoring the overall state of charge of the battery, as well as any drifting that may be taking place in either the end of charge pressure or the end of discharge pressure. The use of as many as four strain gauges and four thermistors per battery is recommended. This permits majority voting between at least three of the operational sensors. The thermistors should be located as close as possible to the warmest portion of the warmest cell. Thermal models can be used to predict this location.

Pressure or onboard ampere-hour integration is recommended as the primary form of charge control, with temperature-compensated voltage used as a hardwired backup charge control mode. Over-temperature protection should always be provided to prevent catastrophic overcharge.

### 2.5.4 Bypass Diodes/Mechanical Switches

The use of bypass diodes, as suggested by SAFT and others, is not universally accepted as a way to bypass a cell that has failed "open" (Refs. 21, 22, 23). The spacecraft's architecture and redundancy provisions determine whether bypass diodes are the most weight-effective way of coping with this power system problem. Typically, bypass diodes, mechanical switches, or thermally activated switches are required where cell level redundancy is selected and are not used

when a battery level redundancy is selected. Diodes not only add weight to the battery system, but they also impact the thermal control considerations of the overall spacecraft. This will be addressed further in the section on redundancy and architecture.

#### 2.5.5 Storage Considerations

There are a number of procedures that are presently used for storing nickel hydrogen cells from the time the cells are activated until the time the spacecraft is put into service. The appropriateness of each procedure is typically dictated by the ease of implementation and the cost of storage, rather than by what is best for the cells. In many cases, the state of precharge in the cells is important in defining acceptable storage conditions. It is generally accepted today that cells or batteries that are either hydrogen precharged or have only a small amount of nickel precharge must be stored with a voltage of over 0.75 V on each cell. In general, and irrespective of precharge type and amount, the preferred storage method is maintenance of a voltage on each cell, preferably without significant overcharge. The following paragraphs outline the different methods of cell storage now in use. Generally, the active or pseudo-active storage modes, such as trickle or top-charge, are at 10°C to 15°C, while the passive storage modes are at 0°C whenever possible. During some test operations, room temperature passive storage may be necessary for limited periods of time.

For nickel hydrogen cells that have an ample amount of nickel precharge (more than 15% of the cell capacity), a discharged and open-circuited storage condition can be used (Refs. 13, 14, 18). In this situation, the nickel precharge holds the desired storage potentials on the cell electrodes. This is the cheapest and simplest storage mode. However, slow reactions occur in the nickel hydrogen cell that, over time, convert the nickel precharge to a less active form. Thus, after a time in discharged storage, a cell will lose its "active" nickel precharge, and will begin to degrade in performance, as would a hydrogen precharged cell. If discharged and open-circuit storage is used, the storage mode must be changed before any cell reaches this effective hydrogen precharged condition, or performance fading will follow. This time period is approximately 1 year for every 4%-5% precharge in the cell at room temperature. At present, it is not clear whether this rate drops significantly at reduced storage temperatures.

Trickle-charge storage is used to maintain a voltage (and significant charge) in all cells during storage. This is a safe storage mode that is independent of type or quantity of precharge. The only drawback is that the internal pressure can slowly climb, due to corrosion of the nickel



electrode sinter structure. Typically, the rate of this corrosion process is slow (and somewhat variable between different lots and designs), requiring many years to become a performance issue.

Top-charge storage is also used to maintain a voltage on each cell, and also is acceptable for all precharge situations. Typically, a cell or battery is charged to a near full state, then allowed to stand open-circuited for 1 to 2 weeks, or a time short enough to guarantee that no cell will fully self-discharge. The cells or batteries are then charged up again, and this sequence is repeated through the storage period. This storage mode has also been associated with some increase in internal cell pressure during storage, but is widely used because the heat dissipation by the battery can be considerably less than in the trickle storage mode, where more significant cooling requirements are usually needed. Periodic rebalancing of cell capacities in a battery may be required with this storage mode.

Controlled voltage at about 1.2 V on discharged cells is the method recommended as the best approach to maintaining cell or battery performance. This approach does not depend on the type and amount of precharge in the cell, and prevents platinum corrosion in the hydrogen electrode during storage of nickel precharged cells. It also minimizes the rate of nickel sinter corrosion while maintaining a safe voltage level on the nickel electrode. This storage method can be easily implemented at the battery or cell level by placing diodes with the desired forward bias across each cell, then applying a trickle charge rate to the string of cells that is high enough to hold all cells at the forward bias of the diodes. Individual leads to measure cell voltage are required in the battery design.

There are a number of operational and test issues that are related to storage. These issues require that nickel hydrogen cells and batteries be prepared for storage in a controlled manner. The first of these general issues is that charged open-circuit stand just before placing cells into storage is to be avoided, since this can result in the temporary isolation of a significant amount of active nickel precharge. Such temporary isolation can be eliminated if cells are prepared for storage by doing a full charge and then an immediate and full discharge and letdown at 22°C as preparation for storage. The second issue is that taking cells or batteries in and out of storage should be minimized, particularly with the passive storage regimes. The third issue is that if batteries have remained for any period of time in active storage, they should not be returned to, or placed into, passive storage unless special tests are performed to verify precharge levels in all cells within the battery. It is possible, and in some situations desirable, to go from passive to active storage, but going from active back to passive storage requires that the precharge status of each cell be verified prior to initiating passive storage.



### 3. Spacecraft Sections

#### 3.1 Redundancy and Architecture

In this section, the following assumptions will be made. (1) The cycle life to be required will be about 14,600 cycles for a 10-year mission. The data base for nickel hydrogen cells suggests that at a DOD of 50%, this requirement would be easily attained. The 50% value is based on a built-in 20% capacity margin beyond the nameplate rating at the beginning of life. This extra 20% is typical of manufactured articles. (2) Only buses resulting from using 22-cell batteries will be examined in detail. Higher voltages, if desired, should be approached using boost regulators. Higher voltage battery strings result in the use of low capacity cells, which have lower energy densities compared to 50 to 70 Ah cell sizes. Further, high voltage strings result in lower reliability numbers for the overall system due to having fewer batteries in parallel. (3) A power level of 5.0 kW at end of life and (4) a minimum voltage per cell of 1.10 V will also be used. The method of battery redundancy to achieve the reliability goal of the spacecraft must also be defined before the battery can be sized.

With cell level redundancy, the vehicle employs only one battery, generally with a boost converter to provide a regulated bus voltage. More cells than initially required are flown in the battery, with bypass diodes and relays around each cell. Large cell sizes are required, greater than 160 Ah, but large capacity cells have been built and tested. Charge control can be accomplished by ampere-hour integration from the ground or by spacecraft computer control. Pressure can also be used for charge control, but the reliability of the strain gauges and strain gauge electronics must be factored into the battery reliability. Because of reliability concerns, pressure charge control is generally backed up with another charge control method, a method that uses a primary battery characteristic such as current (ampere-hour integration) or a temperature-compensated voltage. Within the high radiation environment of this mission, temperature compensated voltage charge control (VT) is recommended as the backup control since it can be hardwired. Over-temperature protection should be included as a hardwired safety backup to prevent a catastrophic overcharge situation.

Redundancy at the battery level is achieved by flying one more battery than is required to meet the mission power and cycle life requirements. Smaller cell sizes are very useful with this approach to redundancy. The power system can be a regulated bus or a battery dominated bus architecture. In the battery dominated architecture, the bus voltage is regulated in the sunlight to prevent overvoltage of the bus, but the bus voltage tracks the battery voltage during discharge. Generally, a battery dominated bus requires simpler electronics than a fully regulated bus, and it has a higher efficiency, particularly when a significant amount of the payload does not require good voltage regulation. With battery level redundancy, bypass diodes are not required. Charge control can be achieved by pressure, ampere-hour integration, or temperature-compensated voltage. Charge control electronics can be simple, with several "primary" charge control methods available. Individual battery charge control is highly desirable. This does require more charge control circuits.

Cell level redundancy saves battery weight over battery level redundancy, even with the additional weight of the bypass diodes and relays. However, cell level redundancy requires more complicated electronics for power system operation than required with battery level redundancy. In general, reconditioning is not an option with cell level redundancy, so it is more difficult to track battery performance trends, and the power and thermal control systems must be designed to operate with nickel memory effect. Given that well-built cells rarely fail "open," a system with battery level redundancy will give longer life than a system with cell level redundancy, due to lower normal operating DODs.

Operating examples exist of both cell and battery redundancy. However, considering the capacity requirements of this application, we recommend battery level redundancy. Sample calculations illustrate the way we approached the cell, battery, and redundancy selections. Table 1 summarizes the seven cases that were examined and presents the results in terms of cell sizes as impacted by the selected DOD with and without one complete battery failure. Sample calculations follow.

Spacecraft Energy Requirements - 5000 W for 45 min (0.75 h) = 3.75 kWh

Assume 22 cell batteries, 1.2 V average cell voltage and 1.1 V cell minimum voltage at end of life

Nominal battery voltage =  $22 \times 1.2 = 26.4$  V

Maximum current =  $5000/24.2 = 206.6$  A

Average current =  $5000/26.4 = 189$  A

Capacity discharged during cycle =  $189 \text{ A} \times 0.75 \text{ h} = 142 \text{ Ah}$

Typical charge current =  $(142 \text{ Ah} \times 1.05 \text{ recharge ratio})/5 \text{ h} = 29.8 \text{ A}$

Maximum charge current =  $(142 \text{ Ah} \times 1.10 \text{ recharge ratio})/5 \text{ h} = 31.2 \text{ A}$

Case 1. Three batteries with two needed

If DOD is 40% with all three batteries on line, cell capacity is  $142 \text{ Ah}/(3 \text{ batteries} \times 0.4) = 118 \text{ Ah}$

Cell size is 118 Ah nameplate

With one battery out -  $142/2 = 71 \text{ Ah}$

DOD =  $71/118 = 60\%$

Case 3. Four batteries with three needed

If DOD is 50% with all four batteries on line, cell capacity is  $142 \text{ Ah}/(4 \text{ batteries} \times 0.5) = 71 \text{ Ah}$

Cell size is 71 Ah nameplate

With one battery out -  $142/3 = 47.3 \text{ Ah}$

DOD =  $47.3/71 = 66.6\%$

Discharge rate (4 batteries) =  $0.67 \text{ C}$

(3 batteries) =  $0.89 \text{ C}$

Charge rate (4 batteries) =  $\text{C}/9.5$

(3 batteries) =  $\text{C}/7.2$

Case 7. Five batteries with four needed

If DOD is 50% with all five batteries on line, cell capacity is  $142 \text{ Ah}/(5 \text{ batteries} \times 0.5) = 56.8 \text{ Ah}$

Cell size is 56.8 Ah nameplate

With one battery out -  $142/4 = 35.5 \text{ Ah}$

DOD =  $35.5/56.8 = 62.5\%$

All three of the preceding cases would be acceptable based on current life cycle test data.

### 3.2. In-flight Management and Control

Using the recommended battery level redundancy, battery telemetry is all that is required to operate the system. If a cell were to fail, the battery with the failed cell would effectively take itself off line if the batteries were connected to the bus with diodes. Additional ground intervention could achieve usable energy from the battery that had one cell that had failed "closed." This would further increase mission life, due to reduced DOD with the remaining "good" batteries.

Table 1. Sample Calculations

Case	Cell Cap.	Cell Cap.	Cell Wgt.	Bat. Fact.	Cells/Bat.	Bat. Wgt*.	Num. Bats.	DOD
Num.	Nom. Plt.	Nominal	Grams	Flange		kg		N N-1
1	118	142	3320	1.3	22	95.0	3	40% - 60%
2	95	114	2750	1.3	22	78.7	3	50% - 75%
3	71	85	2020	1.3	22	57.8	4	50% - 66%
4	65	77	1860	1.3	22	53.2	4	55% - 73%
5	59	71	1700	1.3	22	48.6	4	60% - 80%
6	57	68	1630	1.3	22	46.6	5	50% - 62%
7	52	62	1500	1.3	22	42.9	5	55% - 69%
				Skirt				
1	118	142	3320	1.2	22	87.6	3	40% - 60%
2	95	114	2750	1.2	22	72.6	3	50% - 75%
3	71	85	2020	1.2	22	53.3	4	50% - 66%
4	65	77	1860	1.2	22	49.5	4	55% - 73%
5	59	71	1630	1.2	22	44.9	4	60% - 80%
6	57	68	1630	1.2	22	43.0	5	50% - 62%
7	52	62	1500	1.2	22	39.6	5	55% - 69%

\*Battery weight includes cells, skirts or flanges, cold plate, but not the thermal control subsystem.

Individual batteries can be reconditioned, which permits easy trending of battery performance and increased battery energy efficiency. If reconditioning is to be carried out, it is recommended that the battery be discharged to an average of 0.75 V/cell at a C/50 rate.

## 4. Solar Cells and Arrays

### 4.1. Solar Cell and Array Technologies

The types of solar cells and arrays selected for a particular mission are chosen according to trade-offs between weight, area, and stowed volume, and the overall effect of these parameters on the cost and associated risk to the spacecraft and its mission. This section describes the various types of solar cells and arrays presently available to power system designers. The calculations showing the trade-offs between solar cell and array technologies for this proposed application are discussed in a later section.

The most widely used solar cell material on spacecraft is crystalline silicon. The conversion efficiency of silicon solar cells made in production quantities for space use is in the range of 11.5% to 14.3%. This efficiency is dependent on the processing parameters used in fabricating the cell. The standard, "vanilla" type of cell is 8 mils thick, and has a back surface reflector (BSR) and a dual antireflection coating on the cell surface. This cell is designated as a K4¾ cell by industry standard and is about 12.4% efficient. If, in addition to the above features, the cell has a back surface field (BSF), a greater fraction of the photo-generated carriers is collected, which increases the efficiency to nearly 14%. This cell is designated as a K6¾ cell. However, the K6¾ cell is more susceptible to radiation damage. Therefore, end-of-life (EOL) efficiency of a K6¾ cell may be lower than that of a K4¾, depending on the particular orbit and mission time. Higher efficiency silicon cells are in development stages.

In order to save weight, the thickness of the silicon substrate can be reduced. Thin silicon solar cells have been made with thicknesses ranging from 2.2 to 4 mils. The cost of thin cells increases dramatically over that of 8 mil cells, due to the fragility of the thin cells, which can lead to increased breakage at the glassing and panel-assembly steps. These costs may be more than compensated for at the system level by the lower spacecraft launch weight.

Solar cells made of gallium arsenide (GaAs) are the only other type of solar cell that has been fully flight qualified in the United States. Current GaAs cells for space use are grown on a germanium substrate (5.5 or 8 mils thick) and are made with minimum lot averages of 18.5% conversion efficiency. This is approximately 50% more efficient on a per area basis than silicon cells. The overall cost and weight benefits of this technology for a specific mission must be assessed at the system level.

GaAs/Ge solar cells provide the primary power for the British UOSAT-6 experimental satellite, the Japanese CS3a,b comsats, and the British STRV-1A and 1B experimental flights. The latter two spacecraft were launched in June 1994. The NASA Earth Observing Satellite (EOS-AM), Fast Auroral Snapshot Explorer (FAST), and Tropic Rainfall Measuring Mission (TRMM) satellites, as well as Motorola's Iridium satellites, baseline GaAs power. GaAs/Ge cells have flown or will fly on a number of space experiments such as CRRES (June 1990-October 1991), UOSAT-5 (launched July 1991), PASP+ (launched mid-1994), and ASCOT (to be launched in 1995).

For efficiencies higher than about 19% to be obtained from a mass-produced solar cell, a cell is required that contains more than one junction. Multibandgap (also known as cascade or multijunction) solar cells are being developed by both Applied Solar Energy Corporation and Spectrolab, and are expected to be available in production quantities in 1995 in the 22%-24% efficiency range. The design uses a dual junction gallium indium phosphide/gallium arsenide (GaInP<sub>2</sub>/GaAs) cell. The substrate is a germanium wafer. These multijunction cells are configured as two-terminal devices, which allows for direct substitution into existing panel and power-management designs.

The manufacturing design of the array, in terms of cell size and contact type, depends on the price and vendor. Because lay-down costs are high, usually upwards of \$30 per cell, a large area cell will have a cost advantage over a small area cell. Typical cell sizes used in past designs are 2 x 4 cm<sup>2</sup> and 4 x 4 cm<sup>2</sup>, although larger area cells are becoming available. The trade-off must be made between the cell cost and the final cost of assembling the array. To do this, the true cell

cost must be known. This cost will vary, depending on the vendor, the year, and the size of the order. In the trade study below, the cell costs used are estimates only. Contacts to the cell can be either welded or soldered. Again, the cost will depend on the vendor chosen.

The typical array uses a substrate comprised of two face sheets sandwiching an aluminum honeycomb core. The thickness of the face sheets depends on the strength and stiffness required of the structure. A concept being developed by a number of vendors is the flexible blanket solar array. The panel substrate is one or several sheets of Kapton, usually in 50-75  $\mu\text{m}$  thicknesses. An important advantage to this type of array is that the required stowage volume is small. In addition, because the panels are lightweight, a large-area wing can be supported. A significant disadvantage is that the flex blanket does not provide much shielding for an array that must operate in a high radiation orbit. Thus, this array design would not be appropriate for this orbit. The recommended array for this orbit is a rigid panel array. Face sheets commonly used are made of aluminum, graphite, or Kevlar. Graphite face sheets provide greater stiffness for a given face sheet mass, but at an increased cost. Because the spacecraft will experience over 14,000 thermal cycles, a face sheet made of graphite or Kevlar is recommended, due to its lower coefficient of thermal expansion (CTE) as compared to aluminum. This type of face sheet results in less stress on the cell interconnects, and thus larger sized cells can be used.

#### 4.2 Technology Trades for a Generic MEO Mission

The choice of a given technology will always depend on the specific mission and payloads that need to be supported. The analyses below apply to a wing-mounted solar array design for the proposed mission. Because MEO orbits experience high amounts of radiation, the choice of cell technology becomes limited, as discussed below.

An EOL load requirement of 5.0 kW is the design point for this study. For the following calculation of array power, it is assumed that the loads are the same in eclipse as in the sun. The following equation is used for determining the array power:

$$\eta_R P_s(W) T_s(\min) = P_D(W) T_s(\min) + \left( \frac{1}{\eta_c} \right) \left( \frac{1}{\eta_E} \right) \left( \frac{1}{\eta_{DD}} \right) \left( \frac{P_N(W) T_E(\min)}{\eta_D} \right)$$

The charge efficiency for nickel hydrogen batteries is listed as 0.75 in Table 2, but this could vary from 0.70 to 0.80, depending on the battery temperature during charging and the power system architecture. This results in an estimate for the required array EOL power from 6607 to 6759 W.

Note that  $P_s$ , listed in Table 2, represents the *minimum* power level required from the array at EOL. Effects of beta angle to the sun have not been accounted for. Table 2 defines the variables in the preceding equation.

Table 2. Equation Parameters

Parameter	Definition	Value
$\eta_R$	Power Distribution Loss Factor	0.90
$\eta_c$	Charge Regulator Efficiency	0.99
$\eta_E$	Battery Watt-hr Round Trip Efficiency	0.75
$\eta_D$	Discharge Regulator Efficiency	0.99
$\eta_{DD}$	Discharge Distribution Loss Factor	0.97
$P_D$	Total Load Power in Sun	5000 W
$P_N$	Total Load Power in Eclipse	5000 W
$T_s$	Time in Sun	313.77 minutes (min)
$T_E$	Time in Eclipse (battery discharge time)	45.23 minutes (max)
$P_s$	Required Array Power (end-of-life)	6679 W

For the stated orbit, this angle can be as large as 45 deg or more. Because the spacecraft bus is designed to have a single-axis orientation drive for the solar array, if maximum load power at EOL is required to be provided from the solar array at all beta angles, then the array area would need to be enlarged by over 40%. Alternatively, the battery could supply power during peak load periods. In addition, a power budget margin of 5%-20% is generally added for reliability.



To determine the appropriate area for the array, a calculation has to be made that maximizes the specific power of the array as a function of equivalent cover glass shielding for the solar cell. This is done by first calculating the radiation environment for a MEO orbit. This can be achieved with a commercially available computer program such as Spacerad (Severn Communications Corporation). Then a calculation is made for the equivalent electron fluence to which the cell is exposed as a function of shielding thickness, using the NASA/JPL program EQFLUX. Also calculated is the resultant power degradation that is expected. A spreadsheet calculation then determines the panel weight and area for the various shielding thicknesses. Typical values for substrate density, cell packing factor, and electrical losses have been assumed. Plots of the results for silicon (12.4% efficient, 8 mil thick, K4¾ cell) and GaAs/Ge (18.5% efficient, 5.5 mil thick cell) are shown in Figures 4 and 5. Note that the specific power is at the panel level, and does not include a number of inputs to array weight such as hinges, booms, orientation drives, and slip rings. The results of these calculations will depend on the substrate weight and other design parameters, so that the values for specific power should be taken as an estimate. We have chosen the optimal shielding for the array to be the one that results in the largest specific power. However, if the array size is a critical factor due to launch vehicle considerations, an even thicker amount of shielding could be necessary.

Because a spacecraft in this orbit experiences a high fluence of particle radiation, the optimal shielding is thick, 22 mils for silicon and 27 mils for GaAs/Ge. (Note that the calculation gives the optimal fused silica equivalent shielding thickness, where the density of fused silica is  $2.22 \text{ g/cm}^3$ . If a ceria-doped glass with the usual density of  $2.6 \text{ g/cm}^3$  is used, then the actual cover glass thickness will be smaller. In addition, the cover glass/cell adhesive generally adds at least 1 mil equivalent of fused silica shielding to the cell.) The optimal silicon array experiences a 35.4% loss in power at EOL, and an optimal GaAs/Ge array experiences a 20% loss in power at EOL. Thus, the panel weight is large. Other options examined were thin silicon cells and silicon cells with back surface fields and thus have slightly higher efficiencies than the K4¾ examined here. Thin silicon would save in the range of 30 kg for a 6000 W (EOL) array, but the cost of the cells would offset to a large extent the savings realized by the lower weight. The higher efficiency silicon cells degrade faster than the standard K4¾, so that the optimal specific power is less. A

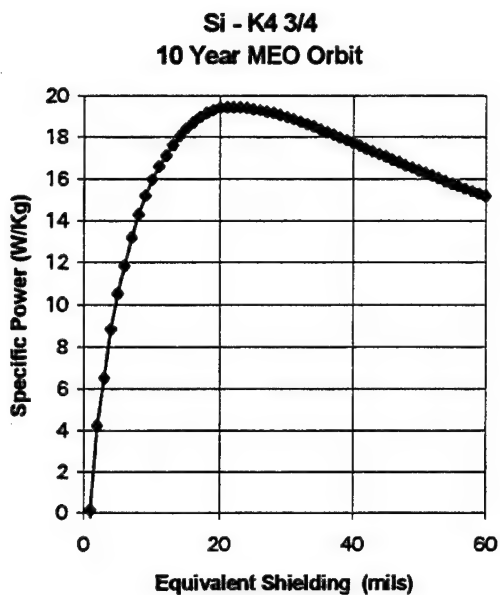


Figure 4. Specific power at the panel level as a function of shielding thickness for a Si solar array in a 10 year MEO orbit.

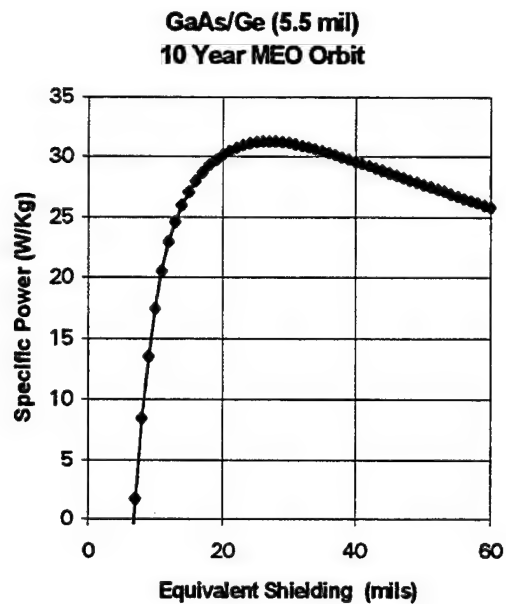


Figure 5. Specific power at the panel level as a function of shielding thickness for a GaAs/Ge solar array in a 10 year MEO orbit.

multibandgap solar cell, GaInP<sub>2</sub>/GaAs, grown on a 5.5-mil Ge substrate, was also included in the trade study. The radiation and temperature performance degradation for the multibandgap cell is assumed to be identical to GaAs/Ge, and thus the optimal shield thickness is the same as well.

Table 3 compares the various costs of silicon and GaAs/Ge rigid solar panels that provide 1000 W of power at EOL during a 10-year MEO mission. At the panel level, these values will scale with power. Flexible blanket arrays were not included in the trade study because the thin shielding they provide for the backside of the cell renders them inappropriate for this orbit. The cover glass listed optimizes specific power. The costs listed in Table 3 should not be taken as exact, since they vary from vendor to vendor and year to year, but as an estimate for comparison purposes only. Larger GaAs/Ge cells, in the range of 20 to 36 cm<sup>2</sup>, are becoming increasingly available, and will most likely be more cost effective for this program. The objective of showing this table is to illustrate how an expensive solar cell can result in lower overall system cost.

The first costs that stands out in Table 3 are the different cell costs. However, while the GaAs/Ge cell costs almost 10 times that of the silicon cell on a per area basis, the cost of a populated GaAs/Ge panel is only about 1/3 more than that of the silicon panel. The difference in panel cost will strongly depend on the substrate cost (materials, design, engineering, and quality assurance testing included), which is estimated to be \$4000/ft<sup>2</sup>. The substrate cost will, of course, vary, depending on the size and complexity of the order. Obviously, the higher the substrate cost, the more favorable the higher efficiency technology. In addition, the weight of the GaAs/Ge panels is 40% lower than that of the silicon panels, and the area is 50% lower. Thus, savings of weight and cost will also arise from the lower amount of structural weight needed to support the

Table 3. Estimated Performance and Costs of Silicon and Advanced Technology Solar Array Panels for a 10-year MEO Mission

<b>EOL POWER = 1 kW</b>	<b>Silicon</b>	<b>GaAs/Ge</b>	<b>Multibandgap (projected)</b>
Efficiency (BOL, 28°C)	12.4	18.5	22
Radiation degradation	0.65	0.80	0.80
Equivalent Fused Silica Shielding Thickness (mils)	22	27	27
Panel weight, kgs (lbs)	51(112)	31(68)	26(57)
Panel area, m <sup>2</sup> (ft <sup>2</sup> )	13.3(143)	6.8(73)	5.8(62)
Panel Specific Power (W/kg)	20	32	39
<b>COSTS</b>			
Cell Area (cm <sup>2</sup> )	24	16	16
Cost per cell*	\$25	\$165	\$220
# Cells required	4990	3853	3240
Cost of cells on panel	\$192K	\$733K	\$812K
Total cost of cell assemblies, including laydown, breakage	\$341K	\$926K	\$975K
Total Panel Cost:** @\$4K/ft <sup>2</sup> substrate cost	\$0.91M	\$1.22M	\$1.22M

\* Cell costs are roughly based on a presentation given by the Applied Solar Energy Corp. at the 1994 Space Power Workshop.

\*\* This is the cost at the panel level only, and does not include the cost of supporting hardware. It is a rough estimate only, and is stated only for the purpose of comparing the costs of the different technologies.

GaAs/Ge panel, as well as reduced amounts of fuel. For example, if a launch cost of \$10,000/lb is assumed, a savings of \$440 K/kW from the reduction in panel weight alone is realized. If one assumes that the weight saved can be utilized to add more channels, then over a 10-year mission, one might generate revenues of \$1 M/kg (Ref. 24).

The production cost for a multibandgap cell is not yet determined. In Table 3, a large differential, a 33% increase over the price of GaAs/Ge cells, is assumed for illustration purposes only. Except for the time required for growing the additional layers, the processing of this type of multibandgap cell is identical to the GaAs/Ge cell. Thus, the only additional factor to affect the yield of this multibandgap cell compared to the GaAs/Ge solar cell processing yield will be the reproducibility of good-quality growths. Even with a 33% increase in cell price, the cost for a GaAs/Ge vs a multibandgap panel is almost the same. Yet the panel area of a multibandgap panel is 16% smaller than that of a GaAs/Ge, and the weight is 16% lower, with a savings of about 5 kg/kW over GaAs/Ge (or 25 kg/kW over silicon). Again, other system cost and weight advantages will also accrue.

Along with cost and weight, a third and important factor is array area. Note that if the supplied power at EOL is around 6.5 kW, then a silicon array would need to provide over 10 kW at beginning of life (BOL). This array would not fit into a medium-sized launch vehicle. A GaAs/Ge or multibandgap solar cell array would need to provide over 8.0 kW at BOL. The power level required from the solar array could increase if full load power were required at the high beta angles, or if a large margin were added for reliability. The area required for a GaAs/Ge array of this size may limit the volume of the spacecraft bus. Thus, if the high efficiency multibandgap cell becomes available in the required time frame and at a price not much in excess of that projected here, then this cell would be highly desirable.

## 5. Summary

### 5.1. The Battery

The study recommends the use of IPV nickel hydrogen cells having the following design features:

1. Operating pressure: as dictated by energy density/efficiency trade-off
2. Axial terminal configuration
3. Back-to-back stacking of components using center core leads
4. Two layers of Zircar separator material
5. Dry powder plaque
6. 26% KOH
7. Catalyzed wall wicks
8. 5/8 in. terminals
9. Strict attention to the most common manufacturing quality issues
10. Strict adherence to suggested storage conditions
11. Use of verified design codes

The study suggests the cells be assembled into batteries with the following characteristics:

1. Operating temperature: +5°C
2. Centrally located heat removal flange
3. Ample monitoring of temperature and pressure integrated in the control features of the spacecraft
4. Battery level redundancy using four batteries where three are required
5. Low battery voltage (28 V), using boost converters for higher bus voltages

The study suggests the adoption of several flight proven battery management features, including:

1. Onboard pressure control with ampere hour integration or pressure control, and VT and T as hard-wired backups.
2. Capability to recondition down to a per cell average voltage of 0.75 V.
3. A DOD selected so that with the loss of one of the four batteries, the cycle life at the new DOD would be in keeping with nominal life cycle test data.
4. Individual charge controllers for the batteries.

Nickel hydrogen cells that would meet these general requirements in the sizes most suited for this application have already been manufactured or are within the design codes of two companies. These cells have already undergone or are still undergoing life cycle testing that has established the confidence for their use in this application.

## 5.2 The Solar Array

The study suggests the use of rigid solar arrays that are populated with gallium arsenide-on-germanium cells with cover glasses that optimize the array size and weight. This recommended cell technology/cover glass combination resulted from a trade-off that factored the initial cell costs, the cost to assemble the cells into arrays, the natural radiation in this orbit, and the required EOL power. A GaAs/Ge solar cell with a 27 mil cover glass results in significant reductions in array size and weight compared with single crystal silicon with its optimum cover glass thickness. The face sheet material recommended for the panel substrates is graphite or Kevlar. The low thermal expansion of these materials allows large area cells to be used in this application.

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